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RESEARCH AND DEVELOPMENT

REPORT 154

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THE AIRCRAFT
STRUCTURAL FACTOR OF SAFETY

by
GEORGE N. MANGURIAN

NOVEMBER 1957



NORTH ATLANTIC TREATY ORGANIZATION
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NORTH ATLANTIC TREATY ORGANIZATION
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THE AIRCRAFT STRUCTURAL FACTOR OF SAFETY

by

George N. Mangurian

SUMMARY

The present aircraft Structural Factor of Safety has been in use for many years. During this period there have been appreciable advancements in structural knowledge; and therefore, a review of the value given to the Aircraft Structural Factor of Safety should be made at this time, and revisions considered wherever deemed possible without jeopardizing any safety that has previously existed.

~~The paper presents~~ certain design aspects wherein a reduction in the presently required ultimate factor of safety is realistic and should be considered as well as other design aspects wherein a reduction should not be considered at this time. A realistic approach must be taken and design philosophies must change if the highest efficiency with adequate safety is to be realized in future aircraft.

SOMMAIRE

La valeur du coefficient de sécurité actuellement employée pour les structures d'avions a été fixée il y a bien des années. Or, la connaissance des structures ayant sensiblement progressé dans l'intervalle, il y a lieu à examiner à nouveau la valeur adoptée pour le coefficient de sécurité des structures d'avions en y apportant toutes modifications utiles compatibles avec la sécurité déjà assurée.

L'auteur de cette Note présente certains aspects du calcul où une réduction de la valeur du coefficient de sécurité ultime couramment exigé est demandée par la réalité et devrait faire l'objet d'une étude; il en expose d'autres où il ne devrait pas être question en ce moment de réduire cette valeur. Il rappelle, en conclusion, la nécessité d'aborder ce problème de façon réaliste et de réorienter nos idées concernant les questions de calcul, pour que les avions de l'avenir réalisent avec une sécurité adéquate le plus grand rendement possible.

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NOTATION

d	actual distance in miles that the airplane travels during its required life
d_0	expected distance in miles that the airplane would have to travel to exceed the ultimate gust velocity at the particular attitude based on the ultimate strength available
F_{tu}	ultimate strength
F_{ty}	yield strength
n	airplane load factor
P_f	probability of failure
U_{de}	limit design gust velocity
U'_{de}	equivalent ultimate gust velocity
V_e	equivalent cruise speed (m.p.h.)
V_t	true cruise speed (m.p.h.)
η	load factor
K.S.I.	denotes kips/in. ² , where a 'kip' is 1000 lb weight

THE AIRCRAFT STRUCTURAL FACTOR OF SAFETY

George N. Mangurian*

1. INTRODUCTION

The present Aircraft Structural Factor of Safety as applied to manned aircraft has been in use for many years. During this period there have been appreciable advancements in structural and aerodynamic knowledge. Therefore, a review of the value given to the Aircraft Structural Factor of Safety should be made at this time; and, if the highest efficiency is to be realized in future aircraft, revisions should be considered wherever deemed possible without jeopardizing the safety that has previously existed.

A paper on this subject was presented by the author¹ some time ago. The present paper reviews some of the points covered in that paper and introduces some additional thoughts on the matter.

2. ULTIMATE FACTOR OF SAFETY

The Ultimate Aircraft Structural Factor of Safety is defined as the ratio of design ultimate load to design limit (or actual applied) load on the structure and is usually equal to 1.5 for U.S. military and commercial aircraft unless otherwise specified. This same value is specified in Amendment No. 85 to the International Standards and Recommended Practices, Airworthiness of Aircraft, Annex 8, I.C.A.O. In the ICAO requirements, Factor of Safety is defined as 'A Design Factor used to provide for the possibility of loads greater than those assumed, and for uncertainties in design and fabrication'. A review of the origin of the 1.5 value for U.S. aircraft, both military and commercial, is given in the earlier paper¹ and it was determined that this value of 1.5 was first established in 1934, and has been with us since that time. It was also determined in this review that the ultimate factor of safety of 1.5 over the limit loads was for the following reasons:-

- (a) Allowance for no permanent set or yielding at limit load
- (b) Allowance for defects in material and workmanship
- (c) Allowance for design uncertainties and inaccuracies
- (d) Allowance for stiffness
- (e) Allowance for exceeding specified maneuvers.

Many of these reasons are similar to those stated in the ICAO requirements.

* Chief Analytical Engineer, Northrop Aircraft, Inc. Northrop Division

Again, Reference 1 discusses these points in considerable detail and points out some of the changes which have come about, or may come about in the future, to influence our thinking with regard to retention of the presently required factor of safety of 1.5. Without repeating this discussion, further thoughts are added here on some of these points.

3. ALLOWANCE FOR NO PERMANENT SET OR YIELDING AT LIMIT LOAD

When the ultimate factor of safety of 1.5 was established for commercial use in 1934 in the United States, the thinking was that an aircraft structure, which was designed so that 1.5 times the limit loads did not exceed the ultimate strength of the materials, would not yield or have permanent set at the limit or applied loading conditions. This was no doubt a reasonable criteria at the time the factor of 1.5 was selected. However, let us take a look at the trend in the use of structural materials since then, to see if this criteria is still realistic. Figure 1 shows this trend plotted against the ultimate to yield strength ratio. It may be interesting to note that the ultimate factor of safety of 1.5 was established in 1934 for commercial use at about the same time that 24ST aluminum alloy, having approximately the same value of ultimate to yield strength ratio, came into general use in aircraft design.

Figure 2 shows the variation in the ratio of the ultimate to yield strength with the ultimate strength of aircraft structural materials at room temperature. Figure 3 shows the variation in the ratio of ultimate to yield strength with temperature. As indicated in these two figures, the presently-used high strength aluminum alloys, high heat treat steels and titanium have a ratio of ultimate to yield strength considerably lower than 1.5. If an ultimate factor of safety of 1.5 is used in design, permanent set or yielding in tension of aircraft components using present day materials will no doubt occur considerably above the limit or applied loadings. This is likewise true for compression, since most modern aircraft have thick plate or sandwich construction. Therefore, if a design is to be predicated on the criteria of permanent set or yielding just above limit load, then the ultimate factor of safety must be based on some criteria other than the ratio of ultimate to yield strength of the structure. Goldman² expressed this same thought in his paper on Safety Factor Requirements for Supersonic Aircraft Structures.

4. ALLOWANCE FOR DEFECTS IN MATERIALS AND WORKMANSHIP

This point seems to have been adequately covered in the earlier paper¹; so no further elaboration is necessary, except to note that Goldman, in Reference 2, gives a qualified endorsement to the author's views. He said "There has been considerable improvement over the years in the workmanship of production aircraft ... it seems rather illogical to think that we will never again have poor workmanship during periods of introduction of new materials or new fabrication techniques. It is logical to expect a smaller degree of trouble as our over-all production knowledge increases."

5. ALLOWANCE FOR DESIGN UNCERTAINTIES AND INACCURACIES

This point can be split up as follows:-

- (a) Aeroelastic effects
- (b) Fatigue
- (c) Flutter
- (d) Dynamic effects
- (e) Structural complexity
- (f) Loading spectra and load distribution
- (g) Aerodynamic heating.

Again, most of these areas are covered in detail in Reference 1. However, some further thoughts may be added to the original discussion on fatigue.

Fatigue has been given considerable emphasis in design of aircraft structures in the past few years. Both the 'fail-safe' design philosophy and the 'safe-life' design philosophy as described in Reference 3 are used in designing for fatigue. Generally, the 'fail-safe' design procedure is used for commercial aircraft, and the 'safe-life' design procedure for military aircraft, because of the shorter life span and higher performance. However, the trend recently in the design of military aircraft is to use the 'fail-safe' design philosophy. This has come about because of the uncertainties in the service life of some of the newer materials and types of construction, such as metal bonding and sandwich structures. Recently, much fatigue and static testing has been carried out to prove the 'fail-safe' concept. The 'fail-safe' strength required for U.S. commercial aircraft is specified in paragraph 4b.270(b) of the Civil Air Regulations, Part 4b. Here it states that 'It shall be shown by analysis and/or tests that catastrophic failure or excessive structural deformation, which could adversely affect the flight characteristics of the airplane, are not probable after fatigue failure or obvious partial failure of a single structural element'. The specified gust velocities result in a structure which, after failure of a single element, has approximately 50% of the original ultimate strength. All the current U.S. transports being presently designed are following the 'fail-safe' philosophy. Furthermore, the tests being performed for many of these aircraft are indicating that there is considerably more strength remaining in the structure after failure of a single component than the required strength. In addition to the 'fail-safe' concept, many of the aircraft companies are specifying replacement of parts after a period of time, before the estimated fatigue life is exceeded. This procedure will undoubtedly be used more and more if minimum weight aircraft are to be designed in the future. With more knowledge being gained on the fatigue problem, it might be realistic to consider a reduction in the present required ultimate factor of safety as some part of this factor has been used in the past to prevent failures. This thought was also expressed by E.R. Shanley in Reference 4.

6. ALLOWANCE FOR STIFFNESS

The various points discussed on this subject in Reference 1 still appear to be reasonable. Admittedly, it will become more and more difficult to design future manned and unmanned aircraft for strength rather than stiffness because of the aerodynamic requirements of thin wing and tail surfaces to reduce drag. Adequate stability and control, as well as freedom from flutter in supersonic aircraft, have required in some cases minimum stiffnesses which are greater than that resulting from strength considerations alone. However, there have been many instances where the stiffness requirements have been minimized by such means as judicious location of masses, favorable positioning of the elastic axis, aerodynamic shape changes, aerodynamic surface reliefs, and selecting types of construction, such as sandwich, that would give the greatest possible stiffness with the available materials. In the early stages of design, it is just as necessary to investigate for stiffness as well as strength, and modifications should be made by the tools just mentioned, if adequate stiffness is not attained. Therefore, it is not satisfactory any longer to say that the ultimate factor of safety of 1.5 would take care of stiffness for future aircraft.

7. ALLOWANCE FOR EXCEEDING SPECIFIED MANEUVERS

The most plausible reason for an ultimate factor of safety of 1.5 seems to be to allow for exceeding specified maneuvers. Goldman² has discussed this point in considerable detail. He states that there have been very few cases where transport or very heavy type bomber aircraft have exceeded their design limits, whereas trainers and fighters exceed the design limits quite frequently. Figure 4 shows typical probability curves for a fighter airplane maneuver spectrum, and illustrates the effect that operational factors have on the spectrum. It is evident that the higher the placard load factor specified to the pilot, the more often any given load factor will be exceeded. On the other hand, the higher this placard load factor, the less often the design limit load factor is exceeded. The reasons given by Goldman to substantiate the need for an allowance for exceeding design maneuver limits are all valid considerations. I would like to reiterate that it should be the designer's aim to develop airplane control characteristics that will provide adequate control as needed for the mission of the airplane; but, at the same time to prevent the pilot from exceeding limit loading conditions by more than a small amount. This can be done by aerodynamic means such as limited size of control surfaces, and also by servo-mechanisms.

The trend is definitely to automatic operations in order to successfully accomplish the mission of modern manned aircraft. Missiles are generally designed for factors of safety considerably less than 1.5, due to the use of automatic controls. High speed military manned aircraft of the future will have navigational and fire control systems that will control the aircraft rather than the pilot. Therefore, there should be no need to design above the limit loading conditions in such cases and the design should be predicated on no permanent set or deformation at the limit loading conditions.

8. ULTIMATE FACTOR OF SAFETY AND ULTIMATE LOADING

The present factor of safety philosophy provides more ultimate strength than necessary in some areas of the aircraft and not enough, or at best an inconsistent amount of strength, in other cases. This results in unsafe conditions in some cases and excessive structural weight in other cases, as effectively illustrated in Reference 1. Some additional examples of this inconsistency are now introduced.

8.1 Gust Criteria

Both military and civil criteria establish a limit gust assumed to act either up or down during lg level flight. The limit gust V-n diagram is thus symmetrical about the lg line, as all available experimental data indicates it should be. However, the application of an ultimate factor of safety of 1.5 to the limit gust loads results in a structure capable of withstanding higher up-gusts than down-gusts. This is illustrated for a low and high wing loading airplane respectively in Figures 5 and 6. In the example of the high wing loading airplane (Fig.5), which has an incremental gust load factor of 1.0, the ultimate gust velocity is twice the limit gust velocity in the up-direction but no more than the limit gust velocity in the down-direction at the critical gust loading condition. However, in this case the maneuver requirements will indirectly provide structural capability for higher negative gust velocities, but the fact remains that the ultimate factor of safety as applied to negative gust requirements do not. In the example of the low wing loading airplane (Fig.6), which has an incremental gust load factor of 6.0 at the critical gust loading condition, the ultimate gust velocity is 1.58 times the limit gust velocity in the up-direction and 1.42 times the limit gust velocity in the down-direction. These examples illustrate two points of inconsistency in the use of an ultimate factor of safety. First, for two airplanes operating in the same degree of turbulence, since the design limit gust velocities are the same, the design ultimate gust velocities are considerably different. Furthermore, the design ultimate gust velocities in the up-direction are considerably different than those in the down-direction, which is contradictory to the fact that gusts are the same in either direction.

8.2 Gust Probability

The mission concept and the loadings encountered during the mission of an airplane have been fairly well accepted for fatigue investigations. It is just as rational to consider the same mission concept and the loadings for static strength investigation. Let us apply this concept to determine the frequency of limit and ultimate gust loadings for a fighter type aircraft having a gust V-n diagram similar to the one in Figure 6, and under the following conditions:-

Cruise Mach number = 0.9

Limit design gust velocity at cruise,
 $U_{dl} \text{ (limit)} = +50 \text{ ft/sec (sea level)}$

Ultimate design gust velocity of cruise from Figure 6
 $U_{du} \text{ (ult)} = +79 \text{ ft/sec (sea level)}$

Ultimate design gust load factor = +10.5

Required airplane life = 5000 hours.

The following two cases will be considered:-

Case 1: 20% of life shall be at 10,000-20,000 ft, and 80% at 40,000-50,000 ft

Case 2: 80% of life shall be at 10,000-20,000 ft, and 20% at 40,000-50,000 ft.

Data on total flight miles to equal or exceed a given gust velocity at various altitudes are shown in Figure 7. These curves were determined for a fighter aircraft flying in rough air turbulence as well as clear air turbulence during part of its mission. They were determined in the same manner as those shown in Reference 5, except that a gust of 50 ft/sec was retained from sea level to 20,000 ft and then reduced linearly to 25 ft/sec at 50,000 ft. The high gust velocity data is questionable, because of the very small amount of test data available in this area. However, these data will be assumed as satisfactory to illustrate the points being discussed. For simplification, it is assumed that all the life of the airplane is spent in the cruise condition and under this assumption and from Figures 6 and 7, the data of Table I are obtained. The numbers in this table would indicate that the airplane in both Cases 1 and 2 would have infinite life before exceeding the ultimate gust velocities which were based on ultimate strength at sea level. However, the limit design gust velocities would be exceeded for the low altitude conditions during the required life of the Case 2 airplane, and almost at the required life of the Case 1 airplane. Admittedly, this is a simplified example, and many things have not been considered: such as, the fleet concept, usage in more turbulent air, etc. The conclusions that someone might arrive at with these data would be that the airplane has excessive ultimate strength when the factor of safety of 1.5 is applied to the limit loads obtained from the specified limit gust velocities. Before such conclusions are seriously acted upon, one must consider the probability approach with the mission concept as perhaps a more rational approach than using an arbitrary value for the ultimate factor of safety on gust loadings.

The probability of failure as applied to gust loadings may be given by

$$P_f = 1 - e^{-d/d_0}$$

where

d = actual distance in miles that the airplane travels during its required life

and

d_0 = expected distance in miles that the airplane would have to travel to exceed the ultimate gust velocity at the particular altitude based on the ultimate strength available.

Taking the fighter discussed previously, the probability of failure due to gust if the airplane were to fly its entire life (5000 hours) at any one altitude is determined as shown in Table II. The probability of failure due to gust at various altitudes as determined in this way is shown in Figure 8 for various percentages

of times spent at specific altitudes. Using these data for the two cases of the fighter the following probabilities result:-

Case 1 (20% at 10-20,000 ft); $P_f = 0.00082$

Case 2 (80% at 10-20,000 ft); $P_f = 0.0032$.

If it is assumed that a probability of $P_f = 0.001$ is an acceptable value for this type of aircraft, it is now possible to determine the ultimate factor of safety necessary to attain this probability. For low values of P_f , it may be assumed that $P_f = d/d_0$ and the two cases are now considered.

Case 1

For $P_f = 0.001 = d/d_0$

$$d = 0.20 \times 3.25 \times 10^6 = 6.50 \times 10^5 \text{ miles}$$

$$d_0 = 6.50 \times 10^5 / 0.001 = 6.50 \times 10^8 \text{ miles}$$

Allowable ultimate gust velocity, $U'_{de} = 102 \text{ ft/sec}$ (Fig. 7)

Limit design load factor	= 7.0	} (From Fig. 6)
Ultimate design load factor	= 10.5	

Incremental load factor (due to ultimate gust), $\Delta L.F. = \frac{102}{105} \times 9.5 = 9.2$

Ultimate gust load factor = $9.2 + 1 = 10.2$

Required ultimate factor of safety = $\frac{10.2}{7} = 1.45$.

Case 2

For $P_f = 0.001 = d/d_0$

$$d = 0.80 \times 3.25 \times 10^6 = 2.60 \times 10^6$$

$$d_0 = 2.60 \times 10^6 / 0.001 = 2.60 \times 10^9$$

Allowable ultimate gust velocity, $U'_{de} = 112 \text{ ft/sec}$ (Fig. 7)

$$\Delta L.F. = \frac{112}{105} \times 9.5 = 10.1$$

Ultimate gust load factor = $10.1 + 1 = 11.1$

Required ultimate factor of safety = $\frac{11.1}{7} = 1.59$.

On the basis of the assumed probability of failure of one in a thousand ($P_f = 0.001$) the present ultimate factor of safety requirement of 1.5 would be satisfactory for the Case 1 airplane but would not be satisfactory for the Case 2 airplane. Whereas, on the basis of the study of total flight hours to exceed the ultimate gust velocity, it appeared that there was sufficient life in both cases.

These illustrations are very simplified cases but do indicate inconsistencies in the present criteria. The probability approach using the mission concept must be exploited further in order to establish consistency with any required ultimate factor of safety. Much more gust frequency data, especially at higher altitudes and higher intensities, is necessary if this approach is to be adopted in gust studies.

8.3 Flutter

Another inconsistency in strength margins is the margin of safety requirement for flutter on military aircraft. The margin of safety for flutter has usually been specified as a speed margin, with a 15% margin being the normally selected value, rather than a 50% value as is specified for ultimate strength above limit strength. This speed margin applies to calculations and wind tunnel flutter model tests, and it is not required that the airplane demonstrate this margin in flight. Because of the often catastrophic nature of flutter, it is only necessary to demonstrate that the airplane is free from flutter throughout its range of speed-altitude flight conditions. Therefore, the actual margin of safety of the aircraft with respect to flutter is seldom known.

The requirement for a 15% speed margin requires that the pertinent natural frequencies be 15% higher than those required for a zero flutter safety margin. For a conventional wing, this would require a 15% increase in the wing torsional frequency, which can be obtained only by a 30% increase in torsional stiffness.

In the era of subsonic flight, the 15% flutter margin was applied to limit dive speed. However, for strength considerations, a speed margin of $\sqrt{0.5}$, or 22%, was required above the limit dive speed based on the ultimate factor of safety of 1.5.

With the advent of transonic aircraft, it has become necessary to consider the margin of safety throughout the flight range, because of the conventional dip in flutter speed near Mach number 1 as is illustrated in Figure 9. This indicates that the minimum margin may occur at some point other than at the limit dive speed, and therefore the present flutter margin of safety requirement should apply throughout the flight range.

It is possible to demonstrate the margin of safety on flutter models in a wind tunnel for sea level conditions by decreasing the stiffness of the model by 30% or by increasing the air density in the tunnel by 30% relative to the equivalent airplane values. However, there is no assurance that such a margin of safety exists on the actual airplane, since it cannot be similarly demonstrated in flight. In fact, an unsafe condition could very well exist if the airplane characteristics were modified slightly during its lifetime. A conclusion that may be drawn from this discussion is that there is one facet of the aircraft design presently working at a 15% margin (flutter consideration) above limit conditions compared to another facet working at a 50% margin (strength consideration).

8.4 Aerodynamic Heating

Inconsistency in the application of the ultimate factor of safety concept to heated aircraft structures is discussed by Goldin in Reference 6 under the section 'Creep and Allowable Ultimate Stress'. Goldin states that 'this concept would impose a drastic load conservatism on top of conservatism for: (a) time-duration of the design load, (b) temperature of the structure at the time the design load occurs, and (c) total life of the aircraft'. He has proposed that the ultimate strength of heated structures be based on the following:-

- i Raise the entire limit-load level by some reasonable factor considerably lower than 1.5, for the estimated temperature and total-life time duration.
- ii Use the creep stress-rupture curves to obtain an allowable ultimate stress.

As in fatigue investigations, the mission concept must be applied in investigating structures subjected to aerodynamic heating. No doubt, variation in the usage of the aircraft will affect the time duration of the design load, the temperatures encountered, as well as the load itself. Realistic tolerances must be considered in establishing the limit design conditions and designing for these conditions. Once this is done, then the ultimate design criteria must be based on something other than 1.5 times the limit load.

9. CONCLUSIONS

The examples given here, and those of Reference 1, point out the inconsistencies in the present ultimate structural factor of safety. Some change must be made in the present factor of safety concept if we are to economically design, build and operate aircraft of the future while retaining adequate safety. As indicated in Reference 1, we can place certain design aspects into two categories:-

- Category A.* Design aspects wherein a reduction in ultimate factor of safety *should not be considered* at this time. In fact, to acquire the required safety the ultimate factor of safety might even have to be increased.
- Category B.* Design aspects wherein a reduction in the ultimate factor of safety *should be considered* at this time. However, structure designed in this category should have no perceptible set or yield at or below the limit loading condition.

In each category, adequate consideration must be given to fatigue requirements, since serviceability and maintainability are affected, as well as safety. Examples in each of these categories are given in the following:

Examples in Category A

- (1) When operational requirements of a new aircraft are not definitely determined, and design maneuvering and loading loads and loading distributions cannot be definitely ascertained 'within small tolerances'.

(2) When positive steps are not taken to prevent exceeding the specified design limit maneuver load factors by a large amount, inadvertently due to undesirable low stick force in pounds per g and unduly light control forces in general.

(3) When adequate experimental data are not available for use in design, and before delivery of aircraft.

(4) When structural behavior due to aerodynamic heating or other phenomena cannot be accurately determined.

(5) When non-linearity in aerodynamic data or structural deflections can be catastrophic if the limit design conditions are exceeded only a small amount. This is especially serious, since many people have thought that an ultimate factor of safety of 1.5 indicates that the airplane strength is good for an ultimate load factor of 50% above the limit load factor.

Examples in Category B

(1) Loadings resulting from ram pressure. These can be ascertained fairly accurately and cannot be exceeded if the airplane stays within its specified speed and altitude design limits. Structure such as intake ducts would be considered under this point.

(2) Loadings from pressurization, such as in pressurized cabins. Such loadings are controlled by a pressure relief valve and cannot be exceeded unless malfunction occurs. It may be more economical to install a dual relief system rather than provide excessive strength in the structure for such malfunctions.

(3) Loadings from hydraulic systems which have relief valves. Here again dual relief systems may be more economical than design for ultimate strength.

(4) Thrust loadings from engines and booster rockets. These loads are determined quite accurately, and therefore, cannot be exceeded.

(5) When loadings are limited due to the buffet boundaries of the airplane. This is rather a questionable item, as it is difficult to determine the magnification of the loadings when the buffet boundaries are reached and even exceeded, before obtaining any flight test information.

(6) When loadings are due to negative pressures approaching absolute vacuum.

(7) When loadings are due to true terminal velocity which cannot be exceeded. Affected structures may be canopies, tail surfaces, inlet ducts and a few others, depending on load distributions.

(8) When loadings are due to hinge moment limitations. Many control surfaces, such as flaps, ailerons, all flying tails, etc., have hinge moment limitations due to the available power of the hydraulic operating cylinder. Therefore, the maximum available hinge moments on these surfaces cannot be exceeded. If adequate tolerance is provided to the center of pressure of the loadings on the surface, then the load is the maximum load possible on the surface. In fact, in many cases over-all wing, tail and fuselage critical loads, and airplane load factors, cannot exceed limit values because of such hinge moment limitations.

(9) When the limit loadings result from maximum control surface deflections. Some design criteria specify maximum control surface deflection, for surfaces such as speed brakes and tabs, at all design flight speeds up to maximum. If good experimental data is available from flight or wind tunnel tests, then it can be stated that the limit loadings cannot be exceeded in such cases.

(10) When practical installations of g-limiters and gust alleviators are available. In such cases, limit loadings cannot be exceeded. However, malfunction of such installations should be taken into consideration. Here again it may be more economical to install a dual system to take care of malfunctions.

(11) When automatic controls are installed on aircraft for maneuvering conditions and the loading conditions will be restricted to the specified maneuver loadings. This method is already in use on missiles, and the criteria of an ultimate factor of safety of less than 1.5 has been accepted practice with considerable success. In order for military aircraft of the future to successfully accomplish their mission, it will no doubt be necessary to rely more and more on automatic controls rather than manual operations. The fire control systems in long range bombers and interceptor fighters must be operated under automatic operations in order to obtain the effectiveness designed in them. Commercial air transportation in the jet period may have to rely almost entirely on automatic navigation in order to obtain the necessary safety. Probability of failure of such automatic controls is a reality, and therefore, dual systems may be required.

(12) When 1.5 times the limit loads encountered from the specified limit gust velocities result in ultimate gust expectancies considerably in excess of the estimated life of the airplane, or the established failure probability. The mission concept must be used in this evaluation, and consistency in up- and down-gusts must be retained.

(13) When 1.5 times the limit loads encountered in maneuvers result in ultimate design loads considerably in excess of ultimate loads based on ultimate design load factors. There should be consistency of strength between various structural components of the aircraft.

(14) When stresses are due to aerodynamic heating, and cannot be exceeded provided that the limit design speed of the aircraft and the rate of temperature rise cannot be exceeded. However, tolerances must be taken into account to parameters such as thermal joint conductance and others affecting thermal stresses.

(15) When load limits have been determined by a flight loads demonstration. If an airplane has been subjected to an extensive flight loads program⁶ and it has been demonstrated that certain loading conditions cannot be exceeded, then it should be possible to take advantage of these load limitations in any future modifications of the airplane.

No doubt there are many more items that can be included in the two categories and should be if there is a real earnest interest in taking a rational approach. Of the many items considered, those that are affected by load factor perhaps would contribute most to a lighter designed airplane. Therefore, the probability approach and the mission concept must be applied to all design conditions.

It has been extremely difficult to change the present design philosophies applied to piloted aircraft. The aircraft engineer might improve his design if he could modify his specified design criteria. However, he is constantly faced with a time factor if a request for deviation involving a change in philosophy is apt to delay progress of the airplane program before resolution can be made of the deviation request. His attitude in general is that, as long as his competitors are bound by the same rules, he does not suffer by comparison; certainly not enough to be a lone crusader and do research that will benefit his competitors as much as himself. Furthermore, it is not realistic to expect any incentive for change to appear from the military or civil authorities, because of the constant fear of degrading the safety that presently exists.

Design philosophies have changed with the advent of pilotless aircraft. Design philosophies must change for piloted aircraft of the future. The key to a motivation for change exists in the increasing costs of our aircraft and missiles and the limited or reduced national budgets with which we are faced. The challenge is open to all. A rational and realistic design philosophy approach must be taken in the future. Only by this means will the aircraft industry be able to produce the safest aircraft at the least cost.

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TABLE I

Estimated Flight Hours to Exceed U_{de} and U'_{de}

	Case 1		Case 2	
Altitude (1,000 ft)	10-20	40-50	10-20	40-50
True cruise speed V_t (m.p.h.)	650	596	650	596
Equivalent cruise speed V_e (m.p.h.)	515	274	515	274
% Time at altitude	20	80	80	20
Miles traveled in 5000 hr	6.1×10^5	2.384×10^6	2.6×10^6	5.96×10^5
Limit design gust velocity U_{de} (ft/sec)	50	29	50	29
Equivalent ultimate gust velocity based on ultimate strength at sea level U'_{de} (ult). (ft/sec)	105	197	105	197
Total flight miles to exceed U_{de} (limit) (From Figure 7)	6.3×10^5	5.0×10^6	6.3×10^5	5.0×10^6
Total flight miles to exceed U'_{de} (ult) (From Figure 7)	9×10^8	∞	9×10^8	∞
Total flight hours to exceed U_{de} (limit)	5150	10,500	1210	41,900
Total flight hours to exceed U'_{de} (ult)	7.38×10^6	∞	1.73×10^6	∞

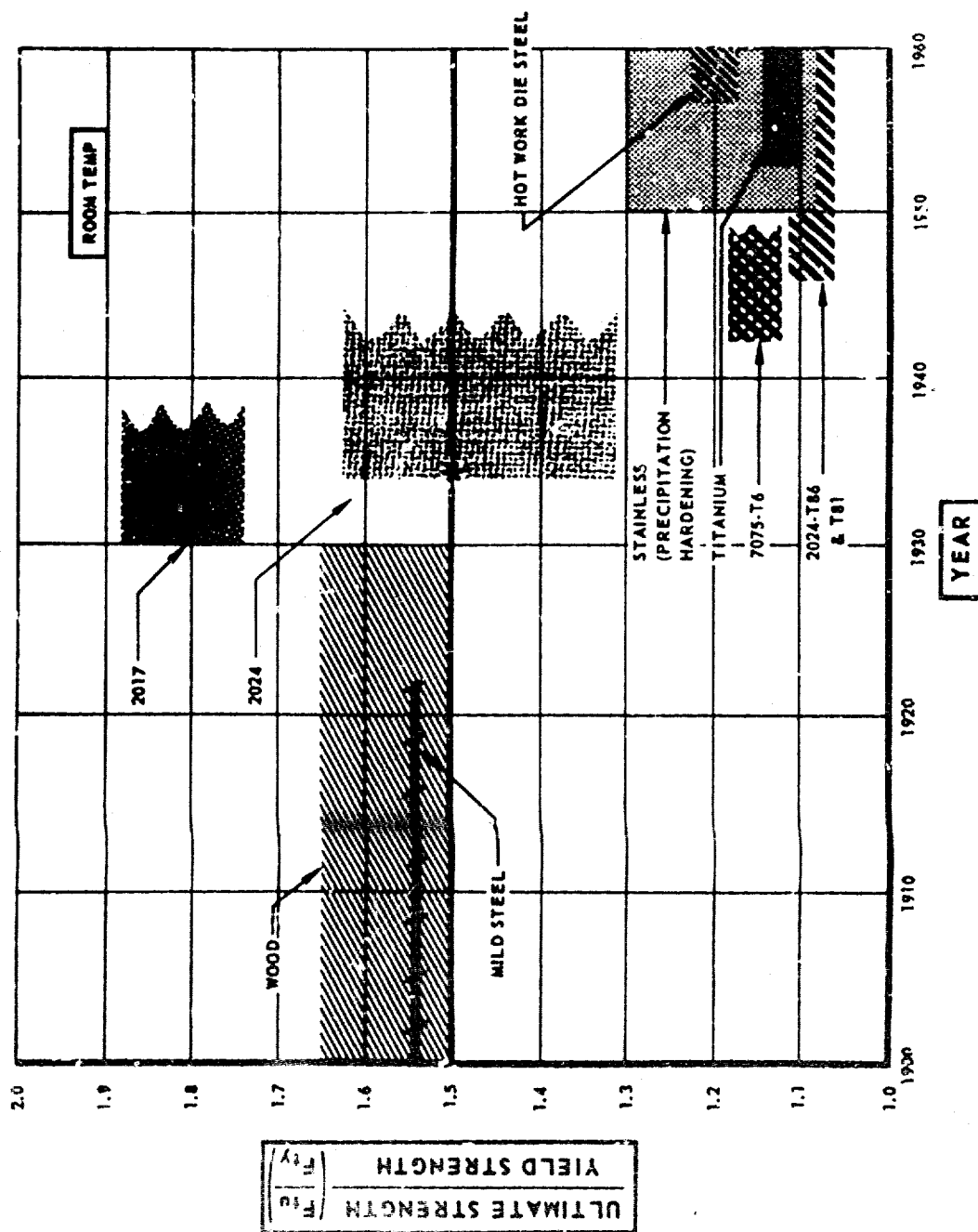


Fig.1 Trend in use of structural materials

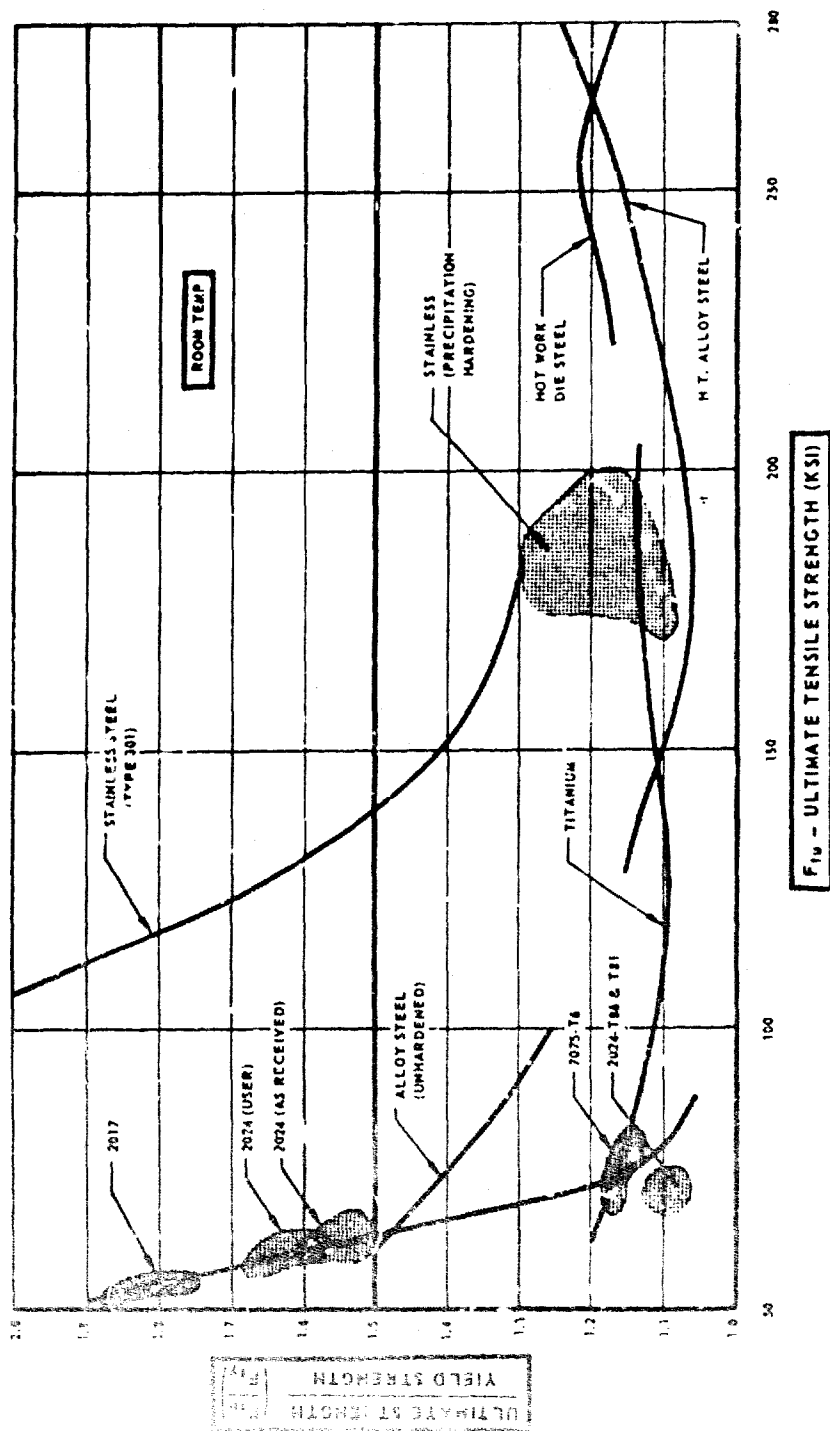


Fig. 2 Variation of P_{tu}/P_{ty} with tensile strength for aircraft structural materials

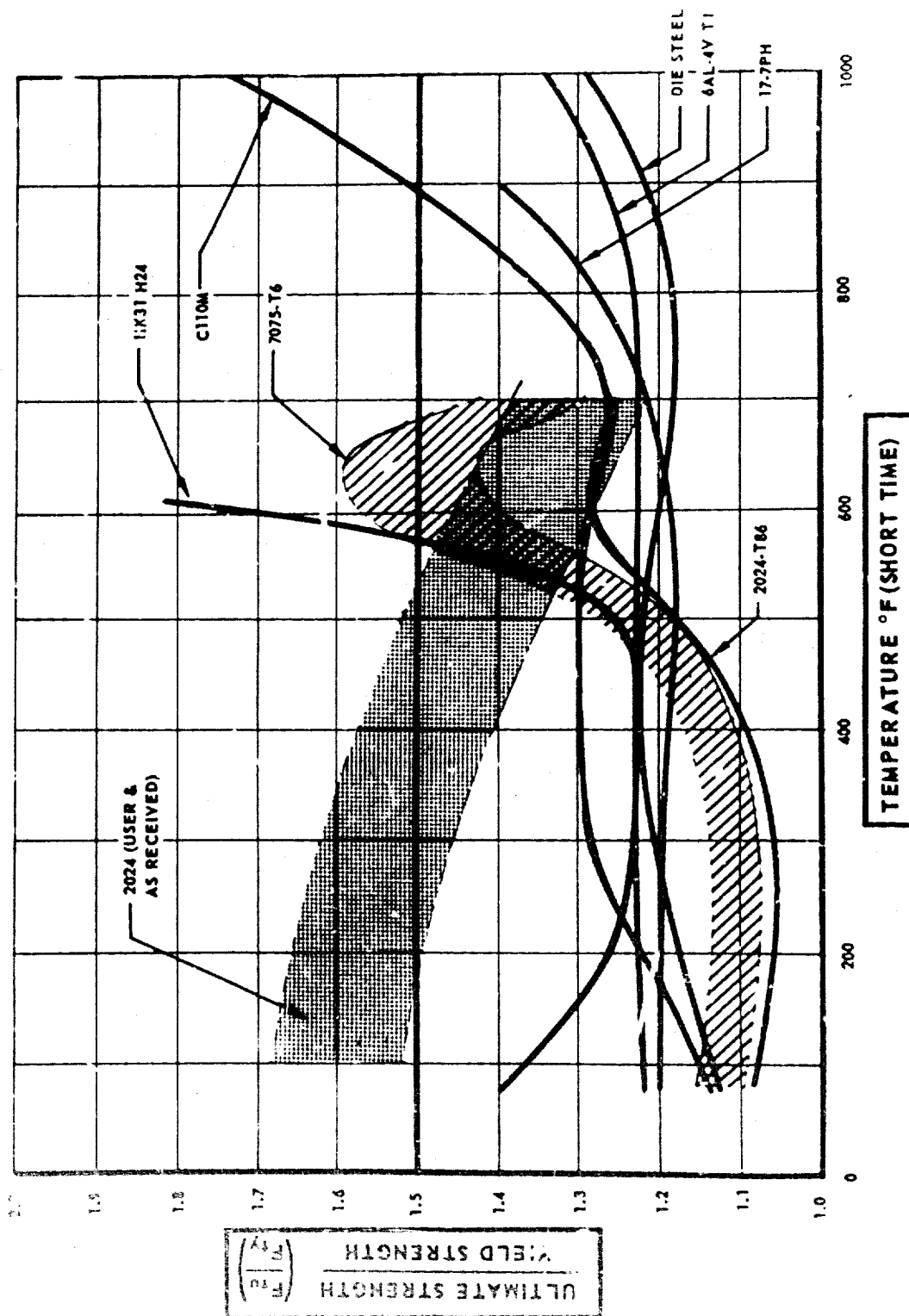


Fig. 3 Variation of F_{tu}/F_{ty} with temperature

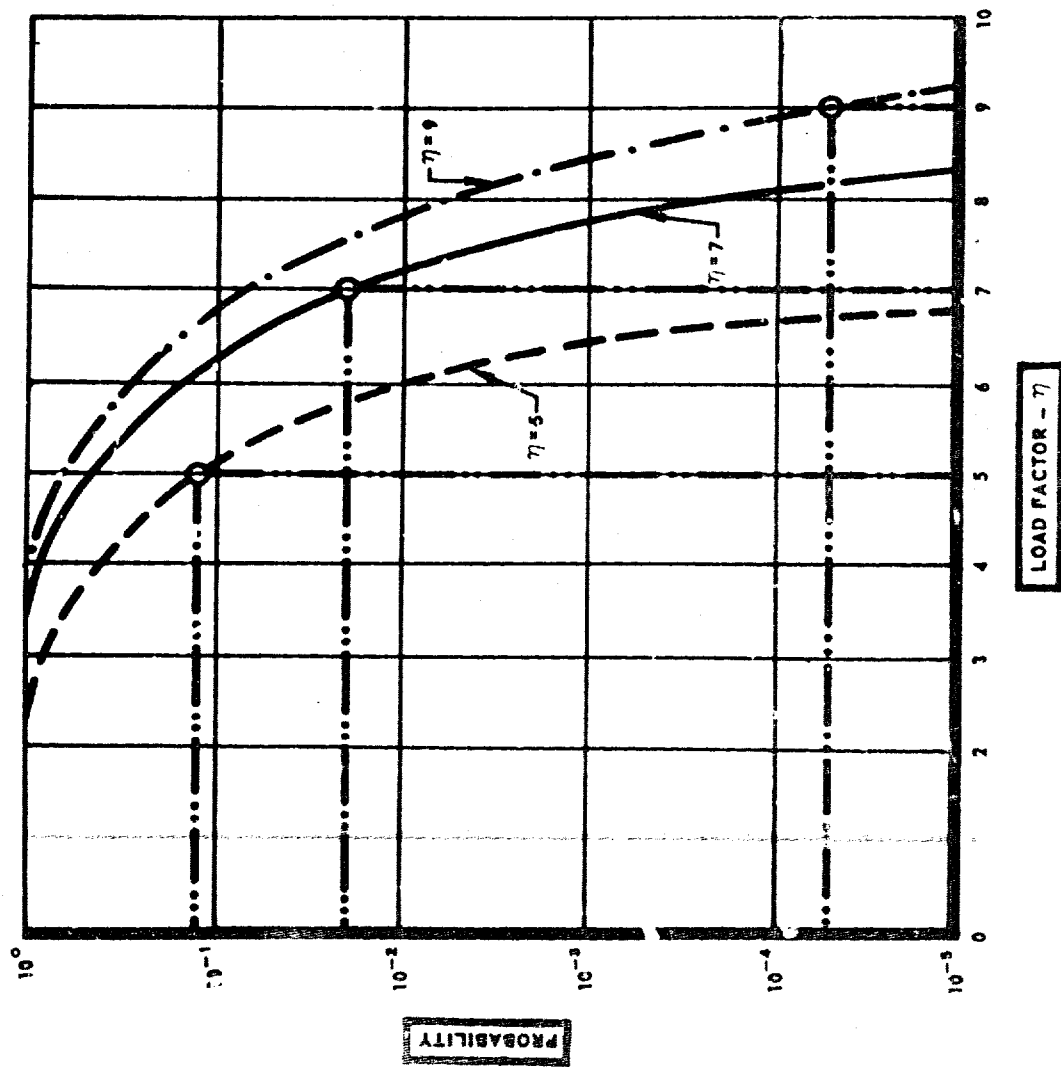


Fig. 4 Probability of exceeding load factor in 'X' hours

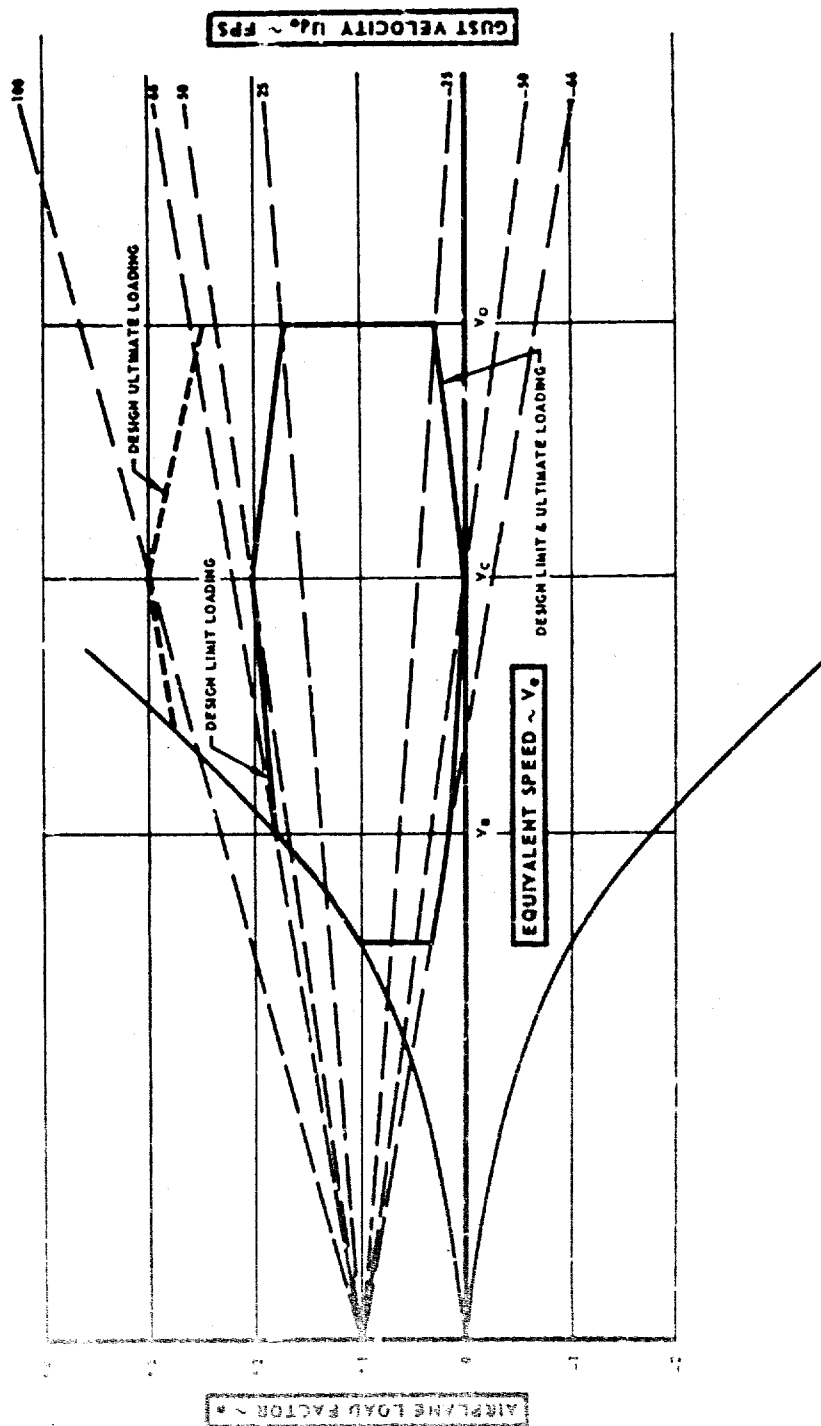


Fig. 5 High wing loading airplane. Typical gust V-n envelope

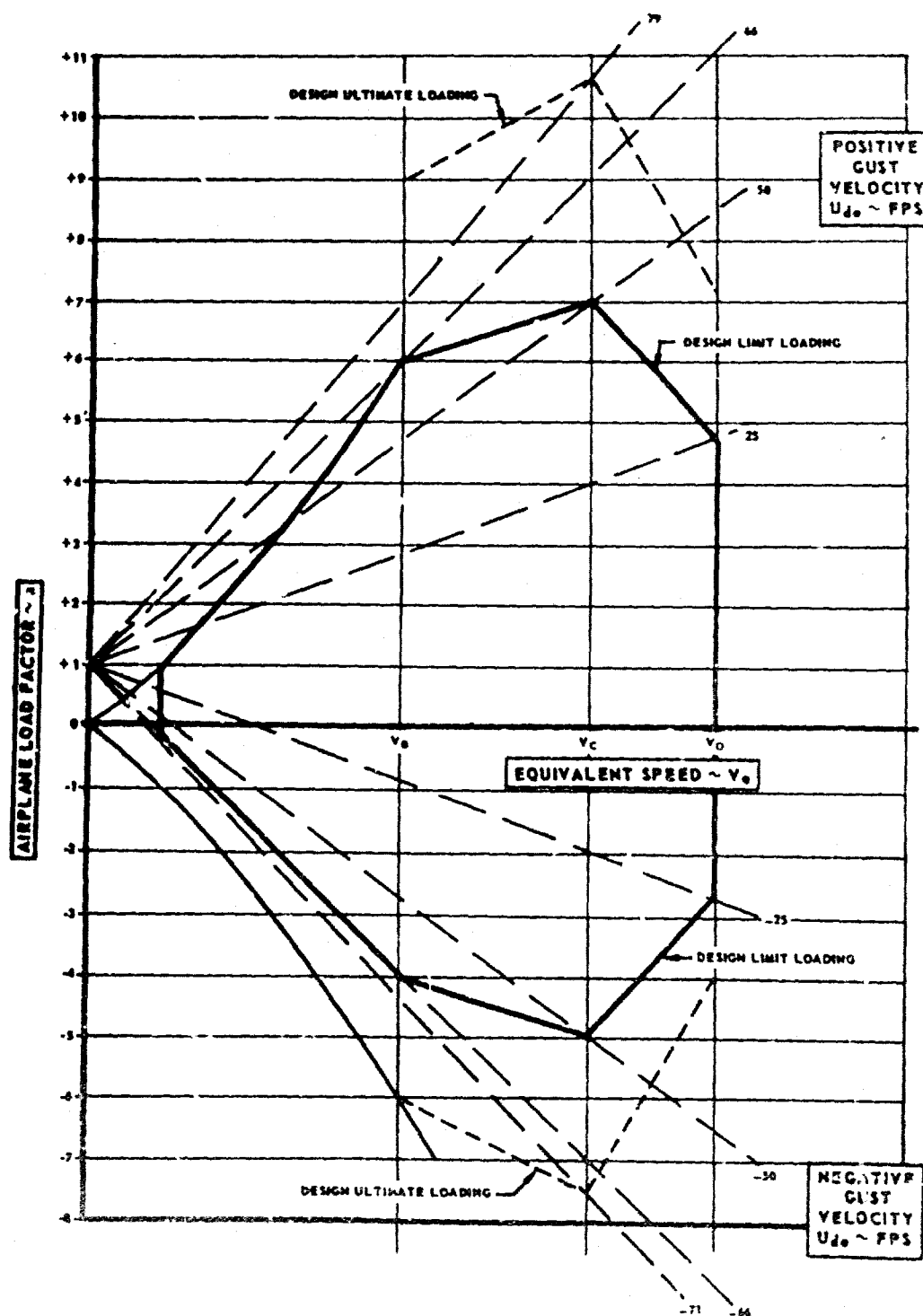


Fig.6 Low wing loading airplane. Typical gust $V-\eta$ diagram

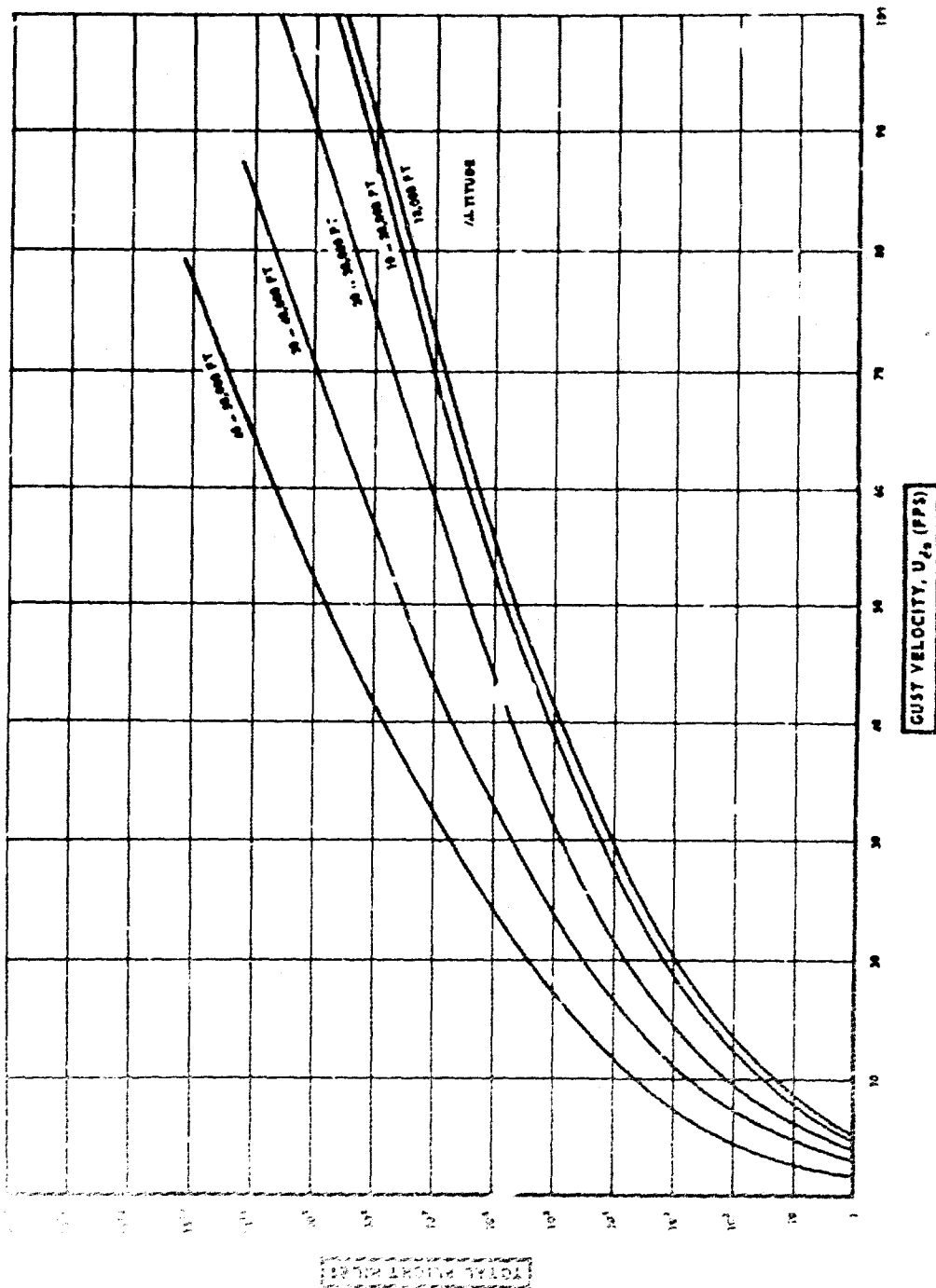


Fig. 7 Total flight miles to equal or exceed U_{de} twice (one cycle)

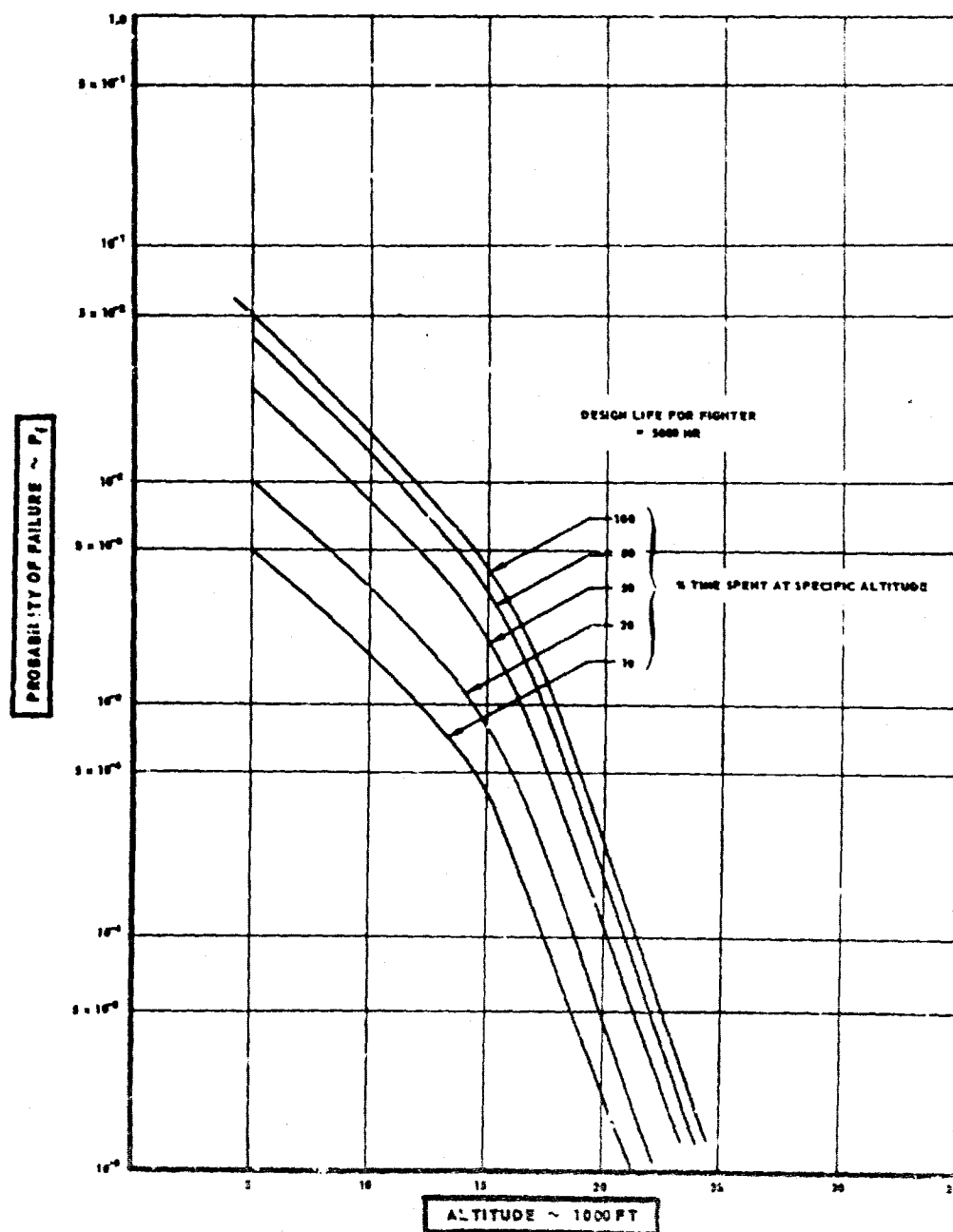


Fig. 8 Probability of failure due to gusts at various altitudes

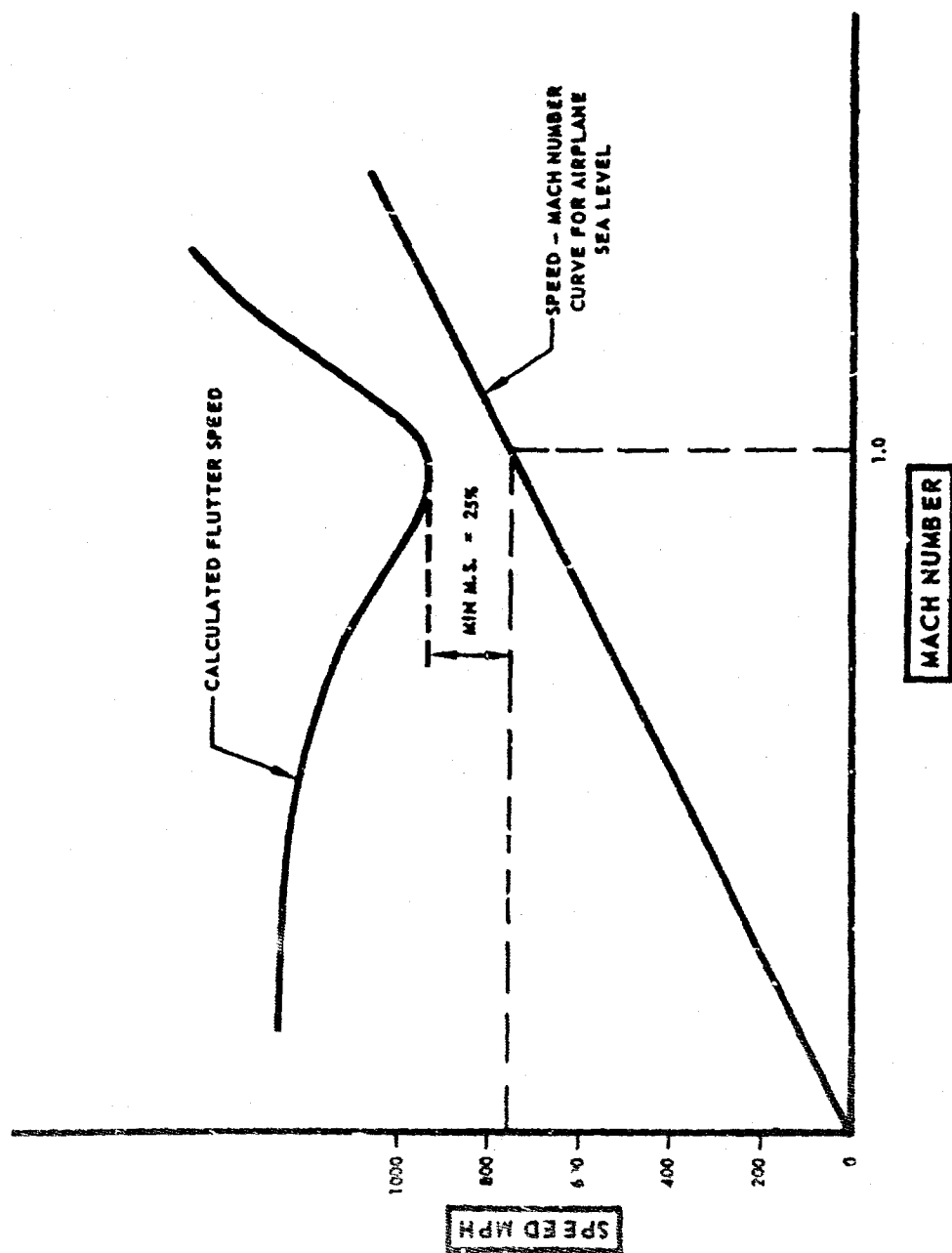


Fig. 9 Flutter speed safety margin versus Mach number

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